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Hydrogen Peroxide Based Propulsion System for Micro Air Vehicle Applications

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Nomenclature

MAV	Micro Air Vehicle
H_2O_2	Hydrogen Peroxide
M	Mach Number
γ	Specific Heat Ratio
m	Mass Flow
η_p	Propulsive Efficiency
rpm	Revolutions Per Minute

Abstract

The miniaturisation of sensors and weapon system will enable the development of micro air vehicles (MAVs) for use in military and surveillance operations, such as signal jamming and intelligence gathering. However, the propulsion systems for MAVs will need to have a range of engine characteristics to satisfy operating requirements.

The Propulsion Department of DERA proposed a hydrogen peroxide (H_2O_2), hybrid rocket/turbine engine for micro air vehicle propulsion. A feasibility study was carried out on the proposed engine and compared against parameters set out in a preliminary specification. The study shows that a bipropellant system with on-board oxygen gives the best flight endurance. The engine configuration will consist of a convergent/divergent nozzle and a ducted fan. This engine can largely meet the physical requirements. It can also meet the indoor and urban reconnaissance requirement of 1-hour flight endurance.

The H_2O_2 engine compares favourably with a rival engine (micro gas turbine). However, the hydrogen peroxide system has an important advantage that it is an established technology and carries considerably less technical risk.

Introduction

It is envisaged that MAVs will have a key role to play in future military and surveillance operations. For these MAVs, it is likely that a range of engine characteristics will be needed to meet specific requirements, such as low speed, low noise, high speed, etc. Features such as weight, ease of starting, reliability, etc. will also feature strongly in the choice of the power plant. Air breathing engines or motors are usually attractive on weight grounds because they do not have to carry their own oxidant. However this may not be so important at small scales when the mass of the engine itself is relatively high. In addition, of course, small engines would have relatively poor thermal and propulsive efficiency due to low cycle temperatures and pressures.

The Propulsion Department of DERA proposed a H_2O_2, hybrid rocket/turbine engine which has a wide range of attractive features. H_2O_2 can, nowadays, be generated 'in the field' by electrolytic techniques. It can be decomposed catalytically to produce steam and oxygen at high temperature and is an acceptable propellant in its own right with a high specific thrust and a low infrared (IR) signature. However since it also contains excess oxidant it would be most efficient to react this with a fuel such as kerosene in a miniature combustion chamber. The high velocity exhaust can be used to drive a turbine coupled to a ducted fan for improving the propulsive efficiency η_p.

Engine Assessment

A systematic study has been undertaken to assess the important operating parameters such as temperature, pressure, weight, efficiency, materials, etc. It also assesses the overall feasibility for an H_2O_2 engine. The physcial and operational requirements for an MAV suitable for military and surveillance applications are define.

Micro Air Vehicle Preliminary Specification

The following specification is provided by the Systems Integration Department of DERA and is based on information supplied by military customers.

Physical requirements -
Aircraft length: 150mm (6.0in)
Gross weight: 100g (3.5oz)
Power plant weight: 50-80g (1.8-2.8oz)
Operational requirements -

Altitude: ground to 60m (200ft)

Short range outdoor reconnaissance: 10-hours endurance, range 32km (20 miles).

Indoor or urban reconnaissance: low speeds, minimum 1-hour endurance.

The above parameters provide the basis for what the power plant has to achieve. Let us first establish how much thrust is needed to keep a 100g aircraft in flight.

Thrust Requirement

The minimum thrust needed to propel a 100g MAV is estimated to be about 0.062N and the corresponding equivalent air speed is about 22m/s **[1]**. The thrust calculations are based upon scaling of a conventional fixed wing aircraft.

Chemical and Physical Characteristics of H_2O_2

Pure hydrogen peroxide is a clear, colourless liquid, of nearly the same viscosity and dielectric constant as water, but of greater density (1442kg/m^3 at 25°C). It is an energy-rich substance that can decompose, yielding steam, oxygen and heat. It can be diluted with water to any concentration. It is insensitive to mechanical shock under normal conditions and can be stored for long periods of time in containers made of suitable materials. If stored in an unsuitable container or if accidentally contaminated with rust or other substances, it can decompose at a rapid rate, releasing large amounts of heat and gas.

H_2O_2 can be decomposed catalytically by many heavy metals (lead, silver, vanadium, etc.) and their salts. A concentration of 90% H_2O_2 is recommended for MAV applications as this concentration is readily available commercially and is much easier to handle and store than pure solutions. Also, reasonable energy is released with the decomposition of this concentration. Moreover, there is the danger of incomplete decomposition for solutions with concentration below 90%.

Equipment and Safety Considerations

Like any high energy substance H_2O_2 requires intelligent care in handling, but given this care it can be used in safety. Materials recommended for handling concentrated H_2O_2 are discussed in **[2]**. For long term storage of H_2O_2, pure aluminium, Teflon and Pyrex glass are suitable. For short-term use, however, equipment can be made of stainless steel or other suitable materials. In deciding which materials should be used for a particular item of equipment, the end use of the item is a dominant factor.

In terms of equipment design, general engineering practices can be considered. For example, the thickness of the wall should be based on pressure requirements. It is also important to include the following:

No place where hydrogen peroxide can be trapped

Valve, pump, etc. must design for complete drainage on shutdown

No dead ends

Avoid dissimilar moving or movable parts

Avoid weld spatter on the inside of equipment

Adopt good machining practice.

For MAV applications, the selection and design of equipment can basically follow the same guidelines as for conventional rockets. However, the equipment system must be made as simple as possible for MAVs due to weight constraint. Possible equipment layout for a monopropellant (H_2O_2 only) system is shown in Figure 1. The system is likely comprised an H_2O_2 storage tank, a solenoid valve and a combustion chamber/nozzle. In addition, a high pressure gas tank (possibly containing nitrogen) and a solenoid valve will be needed to maintain the delivery pressure of H_2O_2. The solenoid valves are used to regulate the flow into the combustion chamber. Electronic circuitry will be needed to drive the valves and this will add weight to the fuel system. Pumps are not recommended as they will increase the complexity and weight of the system. A bursting disc is included in the supply line for safety reason. If 90% H_2O_2 solution is used (will be shown later that the decomposition temperature is about 1022K), stainless steel can be used to construct the combustion chamber/nozzle. The aforementioned items are unlikely to be available commercially as the sizes required are non standard. They will have to be manufactured specifically for MAV applications.

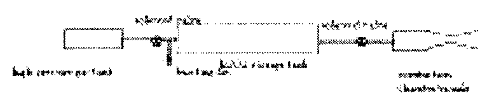

Figure 1 Monopropellant H_2O_2 Fuel System

While for a bipropellant (H_2O_2 and kerosene) system, the equipment selection and design are more complicated. It is inevitable that there will be further weight penalty due to additional equipment for kerosene (storage tank, solenoid valve, etc.). In addition, a high pressure gas tank may be needed to deliver the kerosene fuel to the combustion chamber. The hydrogen peroxide parts will remain the same as a monopropellant system. However, the main problem area will be the design of the combustion chamber. It is shown later that the combustion products will be at high temperatures (circa 2700K). There are few ceramic materials that can withstand such high temperatures. Cooling of the chamber is a possible solution. However, due to the small size of the nozzle the design of cooling system will be very demanding. On the other hand, the ducted fan

configuration to be shown later may alleviate some of the cooling difficulties.

Fuel Tank and Nozzle Parameters

Estimations are performed to obtain the weight of the essential parts of the fuel system [1]. For simplicity, it is assumed that the fuel tank contains 34g of H_2O_2. To hold this weight of fuel, the fuel tank can be a simple cylinder (2cm in diameter and 7.5cm in length). The fuel tank alone will weigh about 16g if it is made of aluminium and its thickness (1mm) should be sufficient to contain the pressure inside the tank. The weight of the combustion chamber/nozzle is estimated to be less than 2g.

H_2O_2 Decomposition

The decomposition of H_2O_2 is an exothermic process in which a substantial rise in temperature occurs. Thermodynamic calculations are carried out on a 90% H_2O_2 solution [1]. The results show that a temperature of 1022K (749°C) and a pressure of 35.5bar (515psi) are achievable when the decomposition products are allowed to expand adiabatically to atmospheric pressure. The results are in agreement with published data [3]. Note that the adiabatic decomposition temperature of pure H_2O_2 is about 1279K (1006°C). Once the temperature and pressure of decomposition are known, it is now possible to assess the flow parameters of the combustion chamber/nozzle.

Combustion Chamber Flow Parameters

A simple convergent/divergent nozzle is used in the flow parameter calculations [1]. Initially, it is assumed that the nozzle has a throat diameter of 1mm and an exit diameter of 3mm. These dimensions are thought to be reasonable and within current manufacturing capability. However, the results show that the mass flow m through the nozzle is too high (3g/s) and the nozzle exit velocity is highly supersonic (M 3.4). Consequently, 34g of H_2O_2 will only last about 11 seconds. Also, the thrust produced (4.42N) is far more than what is required to propel an MAV (0.062N).

In order to reduce m, it is necessary to diminish the combustion chamber pressure and nozzle exit area. A chamber pressure of 2.07bar (30psi) and a nozzle exit diameter of about 2mm will produce a mass flow through the nozzle of about 0.17g/s and an nozzle exit velocity of M 1.1. The thrust produced now is about 0.124N which is comparable to the amount required to propel an MAV. Also, 34g of fuel will last about 3 minutes. It is possible to reduce m further by restricted the supply to the combustion chamber (as discussed later). It is also possible to install a turbine at the nozzle exit and use it to drive a ducted fan. This will have the benefit of increasing η_p of the engine.

Bipropellent Engine

So far, we have only considered the H_2O_2 decomposition process in which oxygen and steam are the end products. Let us now look at the addition of a hydrocarbon fuel to consume the excess oxygen. This may result in an increase in the thrust output of the nozzle. A thermodynamic model based on chemical equilibrium [4] is used to determine the increase in combustion temperature for H_2O_2 and kerosene reaction. The calculations are given in [1]. The results show that 4.7g of kerosene is needed to consume all the oxygen produced by 34g of H_2O_2. The equilibrium temperature is found to be 2768K for this reaction. Although the presence of steam in the reactants will require a certain amount of energy to heat up (effectively lowering the temperature), there is still a large amount of oxygen available for combustion and hence increases the temperature. It is noteworthy that lean air/kerosene combustion has a much lower flame temperature (of the order of 1800K) due to excess nitrogen in air which requires heating and hence lowers the temperature.

Calculations have also been carried out to assess the nozzle flow parameters for a bipropellant engine [1]. It is found that the addition of kerosene has raised the nozzle exit M from 1.1 to 1.15 due to a change in specific heat ratio γ of the combustion products. The nozzle exit temperature is still very high (2423K). However, the mass flow through the nozzle is now reduced to 0.1g/s. The bipropellant fuel (34g + 4.7g = 38.7g) can now support an MAV in flight with the same amount of thrust for over 6 minutes. Therefore, the bipropellant system has a clear advantage in endurance over the monopropellant system. However, the gain in endurance must weigh against the increase in combustion temperature and complexity in the fuel system. At temperatures in excess of 2400K, very few materials will be suitable for making the combustion chamber. Also, very efficient cooling techniques must be implemented to avoid damage to the combustion chamber.

Engine Efficiency

The propulsive efficiency of the nozzle at one of the design conditions is estimated to be about 6%. This is typical of a rocket motor where the jet exit velocity is high and a narrow jet is expelled at the nozzle exit. The thermal efficiency of the nozzle should be high (90% plus) provided there are little heat and pressure losses. The pressure loss through the nozzle should be no more than 1 or 2%. The discharge coefficient of the throat should be better than 0.95 if the nozzle is designed properly. Also, the decomposition of H_2O_2 should be nearly 100% if a suitable catalyst is used.

The propulsive efficiency can be improved significantly (from 6 to 15%) by coupling a turbine

at the nozzle exhaust to drive a ducted fan **[1]**. It should be noted that an ejector is considered to be less practical than a ducted fan due to its relatively low efficiency. The amount of air entrained in the vacuum port will only be a fraction of the core mass flow. While the ducted fan can draw a significantly larger amount of flow provided, there is sufficient momentum in the core flow to drive the turbine.

Ducted Fan Engine

Calculations have been carried out to assess the benefits of a ducted fan engine **[1]**. In the ducted fan engine design, air passes through the outside of the combustion chamber/nozzle (Figure 2). Note that the front of the combustion chamber has been shaped to avoid flow separation. The combustion chamber/nozzle will attain very high temperatures during operation and the bypass flow will help to cool the nozzle. However, a detailed heat transfer study will be required to assess the effectiveness of the cooling air and this has not been pursued further here. For a bypass ratio of 10, the duct exit flow velocity is found to be about 300m/s and the duct exit is 3mm in diameter. The fan rotational speed is estimated to be 1.63E6rpm. This is due to the small size of the fan.

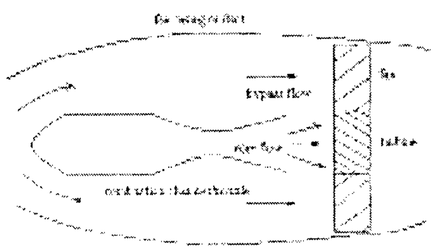

Figure 2 Ducted fan engine

While these calculations are based on a nozzle throat area of 1mm diameter. The total thrust produced by this engine is 0.634N and is significantly more than that required to keep an MAV in flight (0.062N). As a result, the throat diameter can be reduced further but still produces sufficient thrust to power the aircraft. A throat diameter of 0.38mm is found to be sufficient to produce a thrust of 0.062N. With this throat diameter, the flight endurance is now increased to 43 minutes.

Effect of On-board Oxygen

Considerations have been given to the utilisation of an on-board oxygen cylinder as a pressure source for fuel delivery, instead of a nitrogen cylinder which is often suggested in literature **[5]**. If high pressure gaseous oxygen is used, the additional mass due to oxygen (2.4g at 137.93bar) will increase the flight endurance by 2.7 minutes (0.38mm throat). The estimation is based on a storage tank of radius 1cm and length 3cm. On the other hand, if liquid oxygen is used the same volume will increase the endurance by 15.5 minutes. This will give a total flight endurance of 58.5 minutes (43 + 15.5) which is approaching the 1-hour endurance requirement. The penalty of doing this is that the extra oxygen and storage tank will increase the overall equipment weight. Also, liquid oxygen will require refrigeration and insulation which will add weight to the system. A further penalty is that the flame temperature will be higher. Therefore, these factors need to be taken into consideration when choosing the power plant configuration.

Calculations have also been carried out on the performance of an oxygen/kerosene propulsion system **[1]**. If the same amount of fuel is used between O_2/kerosene and H_2O_2/kerosene/oxygen reactions, the flight endurance is about the same due to the same nozzle exit mass flow. However, the flame temperature is even higher (3200K) for O_2/kerosene reaction. Hence even fewer materials will be suitable for making the nozzle.

Discussion

So far, we have considered a variety of H_2O_2 based propulsion systems for MAV applications. The optimum propulsion system would utilise hydrogen peroxide and kerosene as fuel and oxygen as the oxidant. The main results can be summarised as follows:

MAV performance
Minimum thrust - 0.062N
Flight speed - 22m/s
Endurance - nearly 1 hour (based on H_2O_2, kerosene and oxygen)
Engine parameters
Engine weight - 1.6g
Minimum nozzle throat diameter - 0.38mm
Total fuel/oxidant weight - about 52.6g (34g H_2O_2, 7.8g kerosene and 10.8g oxygen)
Total fuel tank weight - about 35g (can be less if necessary)
Main fuel tank dimensions - cylindrical container of 2cm diameter and 7.5cm length for H_2O_2
Preferred engine configuration - combustion chamber/nozzle and ducted fan
Combustion chamber temperature - 2767K
Combustion chamber pressure - 2.07bar
Nozzle exit M - 1.15 (H_2O_2 and kerosene)

The present study has shown that an H_2O_2 based propulsion system can largely meet the physical requirements set out in the preliminary specification. The system can also meet the indoor or urban reconnaissance operational requirement. However, the system cannot meet the short range outdoor reconnaissance requirement (10-hours endurance). In order to meet this requirement, an airframe

similar to a glider may be necessary to keep the MAV in flight. Also, the fuel system may require more complex control functions. It is expected that the altitude requirement of 60m should present no problem as air properties at this altitude are very close to those at sea level where the calculations are based.

Comparison between different engines

A comparison is made among three different types of engine that are small enough and have the potential to power MAVs (Table 1). The bipropellant H_2O_2 engine (with on-board oxygen) is chosen as the baseline for comparison. The micro gas turbine from Massachusetts Institute of Technology MIT [6] is a rival candidate and is shown in Figure 3. The other candidate is a reciprocating diesel engine (Figure 4) from a major model aircraft supplier [7].

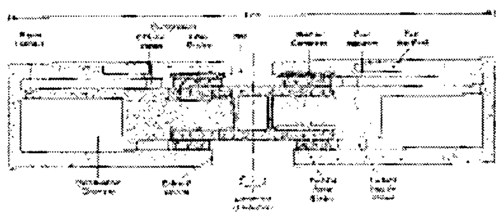

Figure 3 MIT Micro Gas Turbine

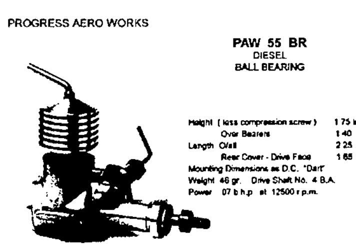

Figure 4 Progress Aero Works Diesel Engine

Operating parameter	H_2O_2 engine	Microturbine	Recip. diesel engine
fuel	H_2O_2/ kerosene	H_2	Paraffin
engine weight	1.6g	2g	46g
mass flow in engine	0.164g/s	0.1 - 0.2g/s	~0.14g/s
fuel consumption	14.9E-3g/s	1.94E-3g/s	~7.94E-3g/s
endurance (same size tank)	58.5min	24.5min	110min
engine rpm	1.63E6 (ducted fan)	2.4E6	14,000
thrust	0.0621N	0.1N	0.065N
thrust/weight ratio	75:1	20-50:1	6:1
engine size	~13mm x φ5mm	~3mm x φ10mm	~44 x 57 x 30mm
combustion efficiency	<90%	60 to 99%	na
overall efficiency	~13%	~17% (claimed)	~11%
exit temperature	~2700K	1500K	~700K
Other parameters			
IR signature	moderate to high	moderate	moderate
noise level	moderate	moderate	High
engine complexity	simple nozzle	Si micromachine	standard
electrical output	no but possible	yes	no

Table 1 Comparison of Operating Parameters between Engine Types

Generally, all three engines seem to confront the same design problems such as high rotational speed, high energy fuel, small combustion volume and low efficiency. It is not unreasonable to expect their performance to be similarly. There are, however, subtle differences in each engine design and consequently one design is significantly better than another. From Table 1, it can be seen that all three engines produce enough thrust to power a 100g MAV, with the H_2O_2 engine having the highest thrust to weight ratio. Both H_2O_2 engine and microturbine have huge weight advantage over diesel engine. In terms of engine size, the diesel engine is also very large by comparison and will have difficulty in fitting into an MAV airframe structure. The diesel engine propulsion system is likely to exceed the MAV weight parameter if fuel, tank and propeller weights are added together. In addition, it is likely to have the highest noise signature due to propeller and piston noises. Judging from size, weight and noise level, the diesel engine is an unlikely candidate to be used in MAV propulsion.

In many ways, the ducted fan H_2O_2 engine is more closely akin to the MIT microturbine than the diesel engine. For example, both H_2O_2 engine and MIT microturbine have very high turbine speed (in excess of a million rpm). Both engines also expect to have similar heat transfer problems. The fuel consumption of the H_2O_2 engine appears to be an order of magnitude greater than the microturbine. One would expect the flight endurance to be much longer for the microturbine. Also, hydrogen has a much higher calorific value than kerosene (3 times) or H_2O_2 (40 times). However, hydrogen is a light substance in either liquid (density ~ 71kg/m^3, boiling

point -253°C) or compressed gas form, it will require a fuel tank much bigger than that for H_2O_2 or kerosene to hold the same weight of fuel **[1]**. In order to fit into an MAV, it is thought that the fuel tank dimensions selected for kerosene and H_2O_2 are reasonable. Therefore, when the same size fuel tank is used for comparison the flight endurance of the microturbine is not quite half the value of the ducted fan engine. Moreover, the H_2O_2 engine has an important advantage over the microturbine, i.e., it is an established technology. Its development will carry considerably less risk and expenditure. It should be noted that gaseous hydrogen is unattractive for MAV applications because of size and weight constraints as a thick wall high pressure container will be needed. On the other hand, refrigeration and thick insulation will be required if liquid hydrogen is used.

<u>Heat loss in small engines</u>

Energy loss due to heat transfer at the walls of the combustion chamber is likely to be significant for small engines. The surface to volume ratio is typically $500m^{-1}$ for micro gas turbine engine and about $125m^{-1}$ for H_2O_2 engine. In a small combustor, it is well known that if the inner diameter of a flame tube is less than some critical size (2 to 3mm), heat transfer from the flame front quenches the reaction. Below this critical diameter, a combustion wave can only be stabilised through external heating of the tube wall. The flammability limits are likely to be affected due to flame quenching. This phenomenon will affect both micro gas turbine and H_2O_2 engine when kerosene fuel is used.

The heat loss to heat released in the combustion process is inversely proportional to the hydraulic diameter of the combustor. The combustion efficiency is therefore unlikely to reach that of a conventional combustor (typically 99.9%) due to significant surface heat losses. Again, this will be true for both microturbine and H_2O_2 engine.

Conclusions

A systematic study has been undertaken to assess the important operating parameters such as temperature, weight, etc. for a H_2O_2 propulsion system. These parameters are evaluated against a preliminary specification drawn up for MAV requirements. The overall feasibility of the system has also been assessed. The main findings are:

1. H_2O_2 is a hazardous substance. Its handling and storage require special attention. A concentration of 90% H_2O_2 is recommended for MAV applications. The propulsion system components are unlikely to be available commercially and will have to be specifically designed and manufactured.
2. A monopropellant (H_2O_2) propulsion system has the advantages of low exhaust temperature and simple equipment design. However, it has the drawback of limited flight endurance.
3. A bipropellant (H_2O_2 and kerosene) propulsion system has a 70% improvement on flight endurance but has high exhaust temperature (circa 2700K) which makes the design and selection of material for the combustion chamber/nozzle very challenging.
4. A bipropellant system with on-board oxygen gives the best flight endurance. The engine configuration will consist of a convergent/divergent nozzle and a ducted fan. This system can largely meet the physical requirements set out in the preliminary specification. It can also meet the indoor and urban reconnaissance requirement of 1-hour flight endurance. However, significant development efforts will be required on the airframe structure and fuel system in order to meet the short range outdoor requirement of 10-hours flight endurance.
5. An H_2O_2 based propulsion system compares favourably with the MIT micro gas turbine. However, the H_2O_2 system has an important advantage that it is an established technology and carries considerably less technical risk. A model aircraft diesel engine is found to be unsuitable for MAV applications on weight, size and noise grounds.
6. Heat loss is expected to be significant in small engines. The combustion efficiency of a hydrogen peroxide based engine will be lower than that of a conventional combustor.

Recommendation

It is proposed that a prototype hydrogen peroxide based engine should be manufactured so that the full benefit of this technology can be established.

Acknowledgement

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